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ESTIMATION OF INCREMENTAL PITCHING MOMENTS DUE TO TRAILING-EDGE FLAPS ON SWEPT AND TRIANGULAR WINGS

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ESTIMATION OF INCREMENTAL PITCHING MOMENTS DUE TO TRAILING-EDGE FLAPS ON SWEPT AND TRIANGULAR WINGS[1]

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SUMMARY

A method is presented whereby incremental pitching moments can be estimated for swept and triangular wings having arbitrary types of trailing-edge high-lift flaps. In the method use is made of span-loading theory together with two-dimensional airfoil data adjusted for the effects of sweep. The method as presented was limited to low speeds and small angles of attack.

Application of the method is demonstrated for some 58 cases covering various types of flaps on wings having a wide range of sweep, aspect ratio, and taper ratio. For all wings, swept as well as triangular, a mean deviation from experiment of about 0.02 in incremental pitching-moment coefficient was found.

Two-dimensional-flap data pertinent to the general application of the method are summarized in graphical form.

INTRODUCTION

The theory of references 1 and 2 permits the rapid determination of the spanwise distribution of lift, lift-curve slope, aerodynamic center, and induced drag for wings having arbitrary plan forms and trailing-edge flap configurations. Calculations of the pitching moment with trailing-edge flaps deflected, however, are outside the scope of this theory since no method of estimating the chordwise distribution of the loading due to flap deflection was included.

The work of reference 3 has demonstrated, on a particular 45° swept-back wing with flaps, how two-dimensional airfoil data and sweep theory can be used to estimate the chordwise load distribution on a swept wing when the spanwise load distribution is known. Once the chordwise and spanwise load distributions are known, of course, the pitching moment can

[1]Supersedes recently declassified NACA RM A55D07 by Harry A. James and Lynn W. Hunton.

readily be determined. The purpose of this report is to present a method for estimating the incremental pitching moment due to trailing-edge flaps on swept and triangular wings by using two-dimensional airfoil data and theory in conjunction with sweep theory. To demonstrate the range of applicability of the procedure, a study has been made wherein measured and estimated pitching moments are compared for a wide variety of flap configurations on swept- and triangular-wing plan forms.

To facilitate the general application of the method, some attention has been given to collecting from numerous sources relevant two-dimensional data for the commonly used types of high-lift flaps, including some data for flaps with area suction or blowing. These results have been summarized herein in graphical form.

NOTATION

A aspect ratio

b wing span

c local chord

$\bar{c}$ mean aerodynamic chord, $\dfrac{\int_0^{b/2} c^2 dy}{\int_0^{b/2} c\ dy}$

C_L lift coefficient, $\dfrac{\text{lift}}{qS}$

C_m pitching-moment coefficient about $\bar{c}/4$, $\dfrac{\text{pitching moment}}{qS\bar{c}}$

c_l section lift coefficient, $\dfrac{\text{section lift}}{qc}$

c_m section pitching-moment coefficient, $\dfrac{\text{section pitching moment}}{qc^2}$

c_{l_α} rate of change of section lift coefficient with angle of attack, per deg

c_{l_δ} rate of change of section lift coefficient with flap deflection, per deg

c_{m_δ} rate of change of section pitching-moment coefficient with flap deflection, per deg

c.p.	center of pressure, percent chord
S	wing area
k	factor equal to $3^{-100(\Delta\eta)^2}$
q	free-stream dynamic pressure
x	longitudinal coordinate from $\bar{c}/4$ to local c.p.
y	lateral coordinate from plane of symmetry
α	angle of attack, deg
α_δ	flap effectiveness parameter, $\frac{c_{l_\delta}}{c_{l_\alpha}}$
Δ	incremental value
δ	angle of flap deflection, measured in plane parallel to plane of symmetry, deg
δ_n	angle of flap deflection for effective section measured in plane normal to the reference sweep line, $\delta_n = \tan^{-1}\left(\frac{\tan \delta}{\cos \Lambda}\right)$, deg
δ'	angle of flap deflection measured in plane normal to hinge line, deg
λ	taper ratio
η	fraction of semispan, $\frac{2y}{b}$
Λ	sweep angle, deg

Subscripts

a	additional lift due to angle of attack
b	basic lift due to camber
f	flap or increment due to flap deflection
av	average

Λ yawed flow

Λ=o two-dimensional or equivalent two-dimensional

METHOD AND APPLICATION

The intent of this report is to supplement references 1 and 2 for the purpose of obtaining estimates of the pitching moment with flaps deflected.[2] The chordwise distribution of the loading due to flap deflection as determined from two-dimensional data are applied, by means of simple-sweep theory, to swept wings and to highly tapered plan forms such as triangular wings.

Section loadings on finite swept wings having moderate taper can rather successfully be related to those in two-dimensional flow through the simple-sweep-theory relations by treating them as untapered wings having a sweep angle equal to that of the $c/4$ line, as demonstrated in references 3 and 4. The primary assumption made is that each section (streamwise) of the finite wing is assumed to behave as that of a yawed infinite wing having identical streamwise geometry and a sweep angle equal to that of the finite wing as illustrated in figure 1. For the yawed infinite wing the chordwise load distributions and centers of pressure of streamwise sections are identical to those of sections normal to the leading edge. These normal sections designated as effective sections can be related directly to two-dimensional airfoil data through simple-sweep-theory relations.

To attempt to apply sweep theory to determine an effective section on wings with large amounts of taper leads to a rather complicated section owing to the variation in sweep angle of the constant-percent-chord lines. Obviously, in the interest of simplicity of application, some approximation is required for this case. Such an approximation is discussed in detail in a subsequent section.

Untapered Swept Wings

A method is developed first for the simpler case involving no taper. One effective section is used for both the additional and basic types of chordwise loading. The following steps are then taken for the purpose of obtaining estimated local centers of pressure.

[2]Another approach to this problem more limited in its applicability is presented in NACA TN 1674 entitled "Estimation of Effectiveness of Flap-Type Controls on Sweptback Wings," 1948, by John G. Lowry and Leslie F. Schneiter.

1. Determine the incremental spanwise load distribution due to flap deflection from available theory such as reference 1 (c_l versus η as shown in fig. 1).

2. Obtain the centers of pressure for the sections (streamwise) of the wing that intersect the flap as follows:

(a) Assume each finite-wing section to be equivalent to one on a yawed infinite wing having a sweep angle equal to that of the finite wing.

(b) Determine the geometry of the effective section on the yawed infinite wing. Being untapered, the flap deflection angle is the only important parameter which differs between the streamwise and effective sections. The flap-chord ratio remains unchanged and the variation in thickness can be ignored.

(c) Solve for an equivalent two-dimensional lift coefficient in unyawed flow for each section that is being considered on the flap of the finite wing.

$$c_{l_{\Lambda=0}} = \frac{c_{l_\Lambda}}{\cos^2\Lambda}$$

where c_{l_Λ} is the incremental lift coefficient due to flap deflection and is equal to c_l.

(d) Determine a center of pressure from two-dimensional airfoil data or from theory of a section having the geometry of the effective section in (b) and at the lift coefficient obtained in (c). Since the section (streamwise) of the finite wing is assumed as identical to that on the infinite wing, the local center of pressure can be assumed to be that found for the effective section.

3. Assume the center of pressure for the unflapped sections of the wing to be located at the 0.25-chord line, except in the regions within 0.20 semispan of the ends of the flap. In this transition region, the center-of-pressure variation can be approximated by the relation c.p. = 0.25 + k(Δc.p.). The value of the constant k and the definition of Δc.p. are given in figure 2. This assumed variation for the center of pressure near the ends of the flap was based primarily on the experimental data shown in figure 3.

4. With the local centers of pressure and the span loading determined, an integration of the section moments about a common axis thus yields the incremental pitching-moment coefficient due to flap deflection

$$C_{m_f} = \frac{-2}{S\bar{c}} \int_0^{b/2} c_{l_\Lambda} cx \, dy$$

Tapered Swept Wings

Introducing taper into the problem rather complicates the determination of an effective section from sweep-theory concepts, owing to the variation in sweep angle of the constant-percent-chord lines. With flaps retracted, the loading is primarily of the additional type and may generally be assumed as concentrated close to the 0.25-chord line. With flaps extended, however, a large portion of the loading is of the basic (camber) type having a much more rearward center of pressure. Since the load line for the additional loading (i.e., quarter-chord line) has been shown (ref. 4) to serve quite satisfactorily as the reference sweep line to define an average effective section for this type of loading, it would then appear reasonable to expect that the basic load line might in similar fashion be used as a reference sweep line to define an effective section for the basic type of loading. Thus, the effect of the varying sweep angle of the constant-percent-chord lines on the chordwise loading can be approximated in a rather simple manner. For the highly tapered wing, two different reference sweep angles become involved in the problem as illustrated in figure 4. Combining these two loads one may derive a local center of pressure as follows:

$$c_{l_\Lambda} = c_{l_{a_{\Lambda=0}}}(\cos^2\Lambda_a) + c_{l_{b_{\Lambda=0}}}(\cos^2\Lambda_b) = c_{l_{a_\Lambda}} + c_{l_{b_\Lambda}}$$

$$c.p._\Lambda = 0.25\left(\frac{c_{l_{a_\Lambda}}}{c_{l_\Lambda}}\right) + c.p._b\left(\frac{c_{l_{b_\Lambda}}}{c_{l_\Lambda}}\right)$$

However, it can be shown that the procedure can be simplified still furthur by use of only the basic load line as the reference sweep line for both components of the loading (additional and basic). Proof that use of only the one load line yields an identical value of $c.p.$ to that found by using both load lines is given in Appendix A. Hence, the more detailed procedure by parts resolves into one no more difficult than that used for untapered wings where only one effective section for both the additional and basic parts of the loading was necessary.

The basic-load reference sweep line required in this method was determined from calculations of center of pressure of the basic load for a plain flap using the section theory of reference 5. In the present analysis the plain-flap theory of this reference has been used for all flap configurations irrespective of the type. This procedure is illustrated in Appendix B and in figure 4. It should be noted that the streamwise geometry of the section considered on the finite wing is identical to that for the fictitious yawed infinite wing; moreover, the effective section is defined on the fictitious yawed infinite wing and not on the finite wing.

Two-Dimensional Data

To facilitate the use of the method, a summary of some pertinent flap parameters and flap data from two-dimensional airfoil tests and theory is given in figure 5. Values of α_δ, c_{l_δ}, $c.p._b$, and c_{m_δ} for a plain flap from the theory of reference 5 are shown in figure 5(a). The values of α_δ and c_{m_f} for various types of flaps given in figures 5(b) and 5(c) were obtained from available test data of references 6 to 54. Use of values of α_δ and c_{m_f} from plain-flap theory are generally applicable for area-suction- and blowing-type flaps employing only sufficient amounts of suction or blowing for maintenance of attached flow on the flaps of the finite wing. Use of these data is demonstrated in Appendix B.

DISCUSSION

A complete summary of the calculations made of the incremental pitching moments due to flaps for some 58 cases on swept and triangular wings at low speed is presented in table I. The measured pitching-moment results for the sample wings were obtained primarily from references 55 to 79. A representative sampling of these results is illustrated in figures 6(a) and 6(b) for the swept and the triangular wings, respectively. Here an attempt has been made to show briefly some results for each of the various types of flap configurations examined. The absolute values of lift and moment indicated in these results were obtained by combining the calculated increments of these quantities with the respective measured values determined from tests of the wing with flaps retracted. The slopes of the estimated pitching-moment curves were determined from the theory of reference 2. An examination of these results shows, surprisingly enough, that little difference in accuracy exists between the swept- and triangular-wing results. An over-all indication of the accuracy of the method for all 58 cases can be seen in the correlation plot of figure 7 where a mean deviation of the order of 0.02 in ΔC_{m_f} was found. The method as presented was limited to the low-speed, small-angle-of-attack range where the longitudinal characteristics are essentially linear, and in the lift range where the loading due to flap deflection can be calculated with reasonably good accuracy. Sample comparisons of measured and estimated span load distributions and local centers of pressure at $\alpha=0^\circ$ are shown in figure 8 for a swept and a triangular wing.

CONCLUDING REMARKS

The low-speed incremental pitching-moment coefficients due to deflection of arbitrary types of partial-span, trailing-edge, high-lift flaps on swept and triangular wings at 0° angle of attack have been estimated and the values correlated with test results for a wide variety of swept- and triangular-wing configurations. The estimates were based on span-loading theory combined with two-dimensional airfoil data corrected to yawed flow conditions.

The results of the study clearly showed that satisfactory estimates of pitching-moment increments could be made for wings with sweepback including those with large amounts of taper such as triangular plan forms. For all wings, the estimated increments of moment coefficient deviated from experiment by a mean value of about 0.02.

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APPENDIX A

USE OF THE BASIC LOAD LINE AS THE REFERENCE SWEEP LINE FOR DETERMINING LOCAL CENTERS OF PRESSURE

The general expression for the loading made up of basic and additional components can be expressed in coefficient form as

$$c_l = c_{l_a} + c_{l_b} \tag{A1}$$

Since the shapes of the loadings are assumed to be invariant with magnitude, the following expression can be used to define local center of pressure for either the finite or two-dimensional case.

$$\text{c.p.} = 0.25\left(\frac{c_{l_a}}{c_l}\right) + \text{c.p.}_b\left(\frac{c_{l_b}}{c_l}\right) \tag{A2}$$

The analysis by parts for the tapered swept wing indicates that

$$c_{l_\Lambda} = c_{l_{a_{\Lambda=0}}}(\cos^2\Lambda_a) + c_{l_{b_{\Lambda=0}}}(\cos^2\Lambda_b) \tag{A3}$$

and

$$\text{c.p.}_\Lambda = 0.25\left(\frac{c_{l_{a_\Lambda}}}{c_{l_\Lambda}}\right) + \text{c.p.}_b\left(\frac{c_{l_{b_\Lambda}}}{c_{l_\Lambda}}\right) \tag{A4}$$

It is the intent now to show that only the value of Λ_b is required in the determination of c.p.

For a particular flap-chord ratio and deflection, $c_{l_{b_{\Lambda=0}}}$ and c.p._b can be determined from theory or two-dimensional data from which the basic loading for the finite wing section can be expressed as

$$c_{l_{b_\Lambda}} = c_{l_{b_{\Lambda=0}}}\cos^2\Lambda_b \tag{A5}$$

which then defines the additional loading

$$c_{l_{a_\Lambda}} = c_{l_\Lambda} - c_{l_{b_\Lambda}} \tag{A6}$$

Substitution of equations (A5) and (A6) into equation (A4) gives

$$c.p._{\Lambda} = \left(\frac{c_{l_{b_{\Lambda=0}}} \cos^2\Lambda_b}{c_{l_{\Lambda}}}\right)(c.p._b - 0.25) + 0.25 \tag{A7}$$

The two-dimensional pitching-moment coefficient may be expressed as

$$-c_{m_{\Lambda=0}} = c_{l_{b_{\Lambda=0}}}(c.p._b - 0.25) \tag{A8}$$

Substitution of equation (A8) into (A7) gives

$$c.p._{\Lambda} = 0.25 - \left(\frac{c_{m_{\Lambda=0}}}{c_{l_{\Lambda}}}\right)\cos^2\Lambda_b \tag{A9}$$

which, it can be seen, does not involve the value of the additional lift reference line Λ_a.

APPENDIX B

SAMPLE CALCULATIONS FOR AN ASPECT-RATIO-2 TRIANGULAR WING WITH A FULL-SPAN, CONSTANT-CHORD, PLAIN FLAP DEFLECTED 10°

(1) η	(2) c_f/c	(3) α_δ	(4) c_{l_Λ}	(5) $c.p._b$	(6) Λ_b, deg	(7) δ_n, deg	(8) c_{m_f}	(9) $c_{l_{\Lambda=0}}$	(10) c.p.
0	0.107	0.39	0.135	0.69	31.0	11.6	-0.1116	0.184	0.86
.1	.112	.42	.143	.69	31.4	11.7	-.1137	.197	.83
.2	.130	.45	.160	.68	32.0	11.8	-.1197	.222	.79
.3	.150	.48	.177	.67	33.3	11.9	-.1255	.257	.74
.4	.175	.52	.203	.66	33.8	12.0	-.1307	.295	.69
.5	.210	.56	.240	.64	34.5	12.1	-.1386	.354	.64
.6	.270	.63	.265	.61	37.0	12.5	-.1450	.416	.60
.7	.360	.70	.313	.57	40.0	13.0	-.1388	.532	.51
.8	.530	.84	.379	.48	46.0	14.3	-.1106	.788	.39
.9	1.000	1.00	.610	---	---	---	0	---	.25
1.0	1.000	1.00	0	---	---	---	0	---	---

(1) Intervals of 0.1 will suffice generally.
(2) From streamwise flap geometry.
(3) Theoretical values from figure 5(a).
(4) Incremental span load distribution due to flap deflection. from available methods such as reference 1.
(5) Plain-flap basic load c.p. from figure 5(a).
(6) Sweep of the constant-percent line through $c.p._b$ from (5).
(7) $\delta_n = \tan^{-1}(\tan \delta/\cos \Lambda_b) = \tan^{-1}(\tan 10^\circ/\cos$ (6)$)$.
(8) From two-dimensional data or theory (theory used in this case), such as in figure 5(a), for c_f/c in (2) and flap deflection δ_n in (7).
(9) $c_{l_{\Lambda=0}} = c_{l_\Lambda}/\cos^2\Lambda_b =$ (4)$/\cos^2$ (6).
(10) From two-dimensional data or theory at $c_{l_{\Lambda=0}}$ from (9), for c_f/c in (2), and flap deflection δ_n in (7); or computed by c.p. $= 0.25 - (c_{m_f}/c_{l_{\Lambda=0}}) = 0.25 -$ (8)/(9).

At this point, several of the accepted procedures may be used with the above information to obtain an incremental pitching-moment coefficient due to flap deflection. The relation $C_{m_f} = \int_0^{1.0} c_{m_{\bar{c}/4}}\left(\frac{c^2}{\bar{c}c_{av}}\right)d\left(\frac{2y}{b}\right)$ from reference 80 is sometimes used; or more simply

$$C_{m_f} = \frac{-2}{S\bar{c}} \int_0^{b/2} c_{l_\Lambda} cx \, dy = -C_{L_f} \frac{x}{\bar{c}}$$

(x is the distance to $\bar{c}/4$ from the wing center of pressure) which for the above example was found to be:

$$C_{m_f} = -0.206\left(\frac{7.90}{16.67}\right) = -0.098$$

Configurations having constant-percent-chord flaps naturally have singular values of (2), (3), (5), (6), (7), and (8) and, consequently, the computations are reduced considerably.

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TABLE I.- SUMMARY OF RESULTS OF MEASURED AND ESTIMATED FLAP PITCHING-MOMENT INCREMENTS

Case No.	A	λ	$\Lambda_{c/4}$	Type flap	Flap extent, 2y/b	c_f/c	δ'	C_{m_f}	
								Measured	Estimated
1	6.0	0.50	45	double-slotted	0.18 to 0.58	0.25	55	-0.14	-0.13
2	3.5	.45	60	double-slotted	.21 to .57	.25	55	-.12	-.09
3	3.5	.30	45	double-slotted	.16 to .70	.25	55	-.30	-.30
4	4.8	.51	35	area-suction	.15 to .50	0.22 to .26	55	-.11	-.12
5	5.1	.38	45	single-slotted	.14 to .45	.25	20	-.04	-.04
6	5.1	.38	45	single-slotted	.14 to .45	.25	30	-.06	-.06
7	5.1	.38	45	single-slotted	.14 to .45	.25	40	-.07	-.07
8	5.1	.38	45	double-slotted	.14 to .45	.25	30	-.07	-.06
9	5.1	.38	45	double-slotted	.14 to .45	.25	40	-.09	-.08
10	5.1	.38	45	double-slotted	.14 to .45	.25	55	-.11	-.12
11	6.0	.50	35	split	.02 to .50	.20	60	-.03	-.03
12	6.0	.50	35	double-slotted	.02 to .50	.25	49	-.13	-.14
13	8.0	.45	45	split	0 to .50	.20	60	.07	.07
14	8.0	.45	45	split	0 to .60	.20	60	.02	.03
15	3.8	.59	47	plain	.10 to .58	.25	30	-.04	-.03
16	3.8	.59	47	plain	.10 to .58	.25	45	-.05	-.03
17	3.8	.59	47	plain	.10 to .58	.25	60	-.05	-.07
18	3.5	.50	45	plain	0 to .50	.20	20	-.04	-.04
19	3.5	.50	45	plain	0 to .50	.20	40	-.06	-.04
20	3.5	.50	45	plain	0 to .50	.20	60	-.07	-.05
21	2.5	.42	40	plain	.20 to .50	.20	53	-.05	-.04
22	9.0	.40	0	split	0 to .60	.20	60	-.16	-.17
23	9.0	.40	0	split	0 to .98	.20	60	-.19	-.20
24	9.0	.40	0	single-slotted	0 to .60	.25	45	-.34	-.35
25	9.0	.40	0	single-slotted	0 to .98	.25	45	-.41	-.42
26	9.0	.40	0	double-slotted	0 to .60	.25	50	-.50	-.50
27	9.0	.40	0	double-slotted	0 to .98	.25	50	-.60	-.61
28	10.0	.40	40	split	.07 to .46	.20	30	.01	-.02
29	10.0	.40	40	split	.07 to .46	.20	60	.04	0
30	2.6	.41	60	split	.07 to .50	.20	60	-.03	-.03
31	3.4	.44	48	plain	.07 to .59	.20	20	-.04	-.02
32	3.4	.44	48	plain	.07 to .59	.20	60	-.06	-.03
33	3.4	.44	48	plain	.07 to .99	.20	20	-.07	-.06
34	3.4	.44	48	plain	.07 to .99	.20	60	-.13	-.12
35	2.0	0	56	single-slotted	0 to .70	.21	40	-.28	-.27
36	2.0	0	45	plain	.18 to 1.00	.25	20	-.14	-.15
37	3.7	.40	44	area-suction	.16 to .50	.22	61	-.16	-.14
38	3.7	.40	44	area-suction	.16 to .75	.22	61	-.28	-.27
39	2.0	0	56	area-suction	.17 to .72	.11 to .33	59	-.29	-.34
40	2.0	0	56	single-slotted	.18 to .70	.21	40	-.23	-.21
41	4.0	0	37	single-slotted	.13 to .67	.13 to .33	40	-.25	-.23
42	2.3	0	52	split	.08 to .67	.11 to .32	49	-.15	-.13
43	2.3	0	52	plain	.08 to .67	.11 to .32	53	-.16	-.13
44	2.3	0	52	double-slotted	.08 to .67	.11 to .32	50	-.32	-.29
45	2.0	0	56	plain	0 to 1.00	.11 to 1.00	10	-.10	-.10
46	2.0	0	53	plain	.12 to 1.00	.13 to 1.00	-10	.10	.09
47	2.3	0	52	plain	0 to .50	.13 to .25	20	-.08	-.09
48	2.3	0	52	plain	0 to .50	.13 to .25	40	-.13	-.13
49	2.3	0	52	plain	0 to 1.00	.13 to 1.00	10	-.08	-.09
50	2.3	0	52	plain	0 to 1.00	.13 to 1.00	20	-.15	-.15
51	2.3	0	52	plain	0 to 1.00	.13 to 1.00	30	-.22	-.18
52	2.0	.20	45	single-slotted	.18 to .96	.11 to .41	40	-.26	-.24
53	2.0	.33	37	single-slotted	.19 to 1.00	.13 to .33	40	-.26	-.26
54	3.0	0	45	single-slotted	.15 to .77	.13 to .50	40	-.15	-.18
55	3.0	.40	16	area-suction	.15 to .75	.29	60	-.29	-.27
56	4.8	.51	35	blowing	.14 to .50	.23	45	-.12	-.13
57	4.8	.51	35	blowing	.14 to .50	.23	60	-.16	-.17
58	2.0	0	56	plain	0 to .70	.21	22	-.14	-.12

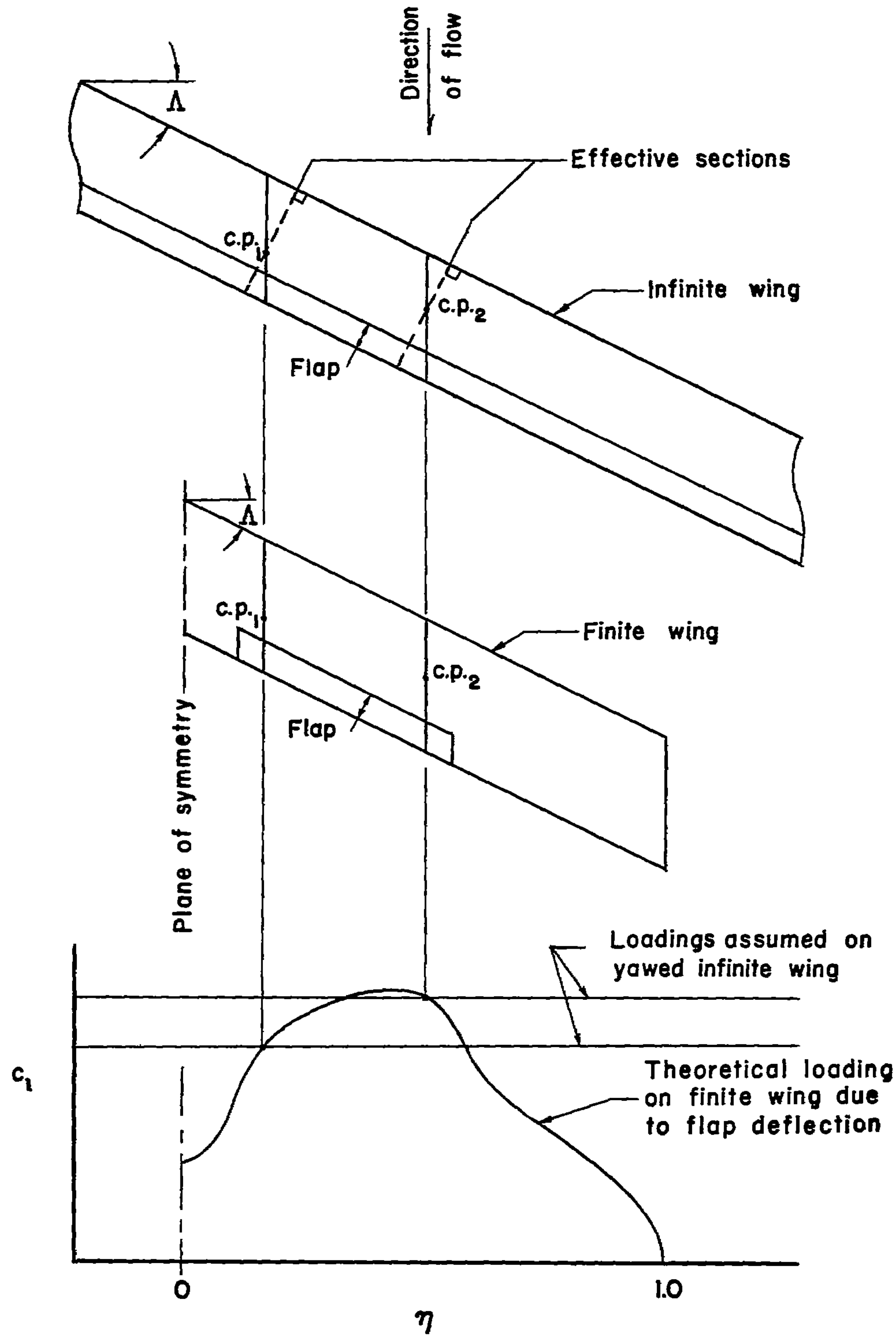
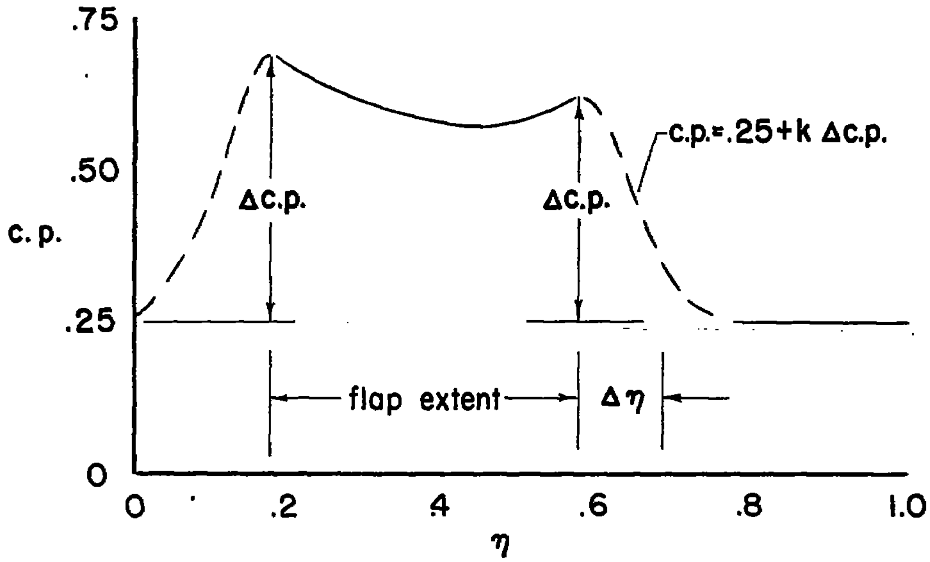


Figure 1.- Theoretical loading for untapered wing with trailing-edge flap.



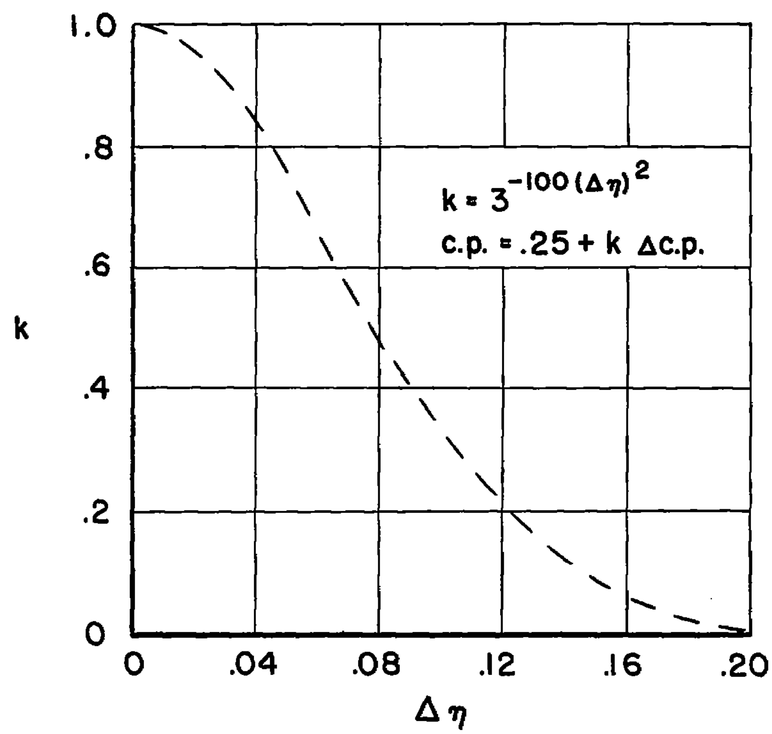


Figure 2.- Method of estimating center of pressure on the unflapped sections near ends of flaps.

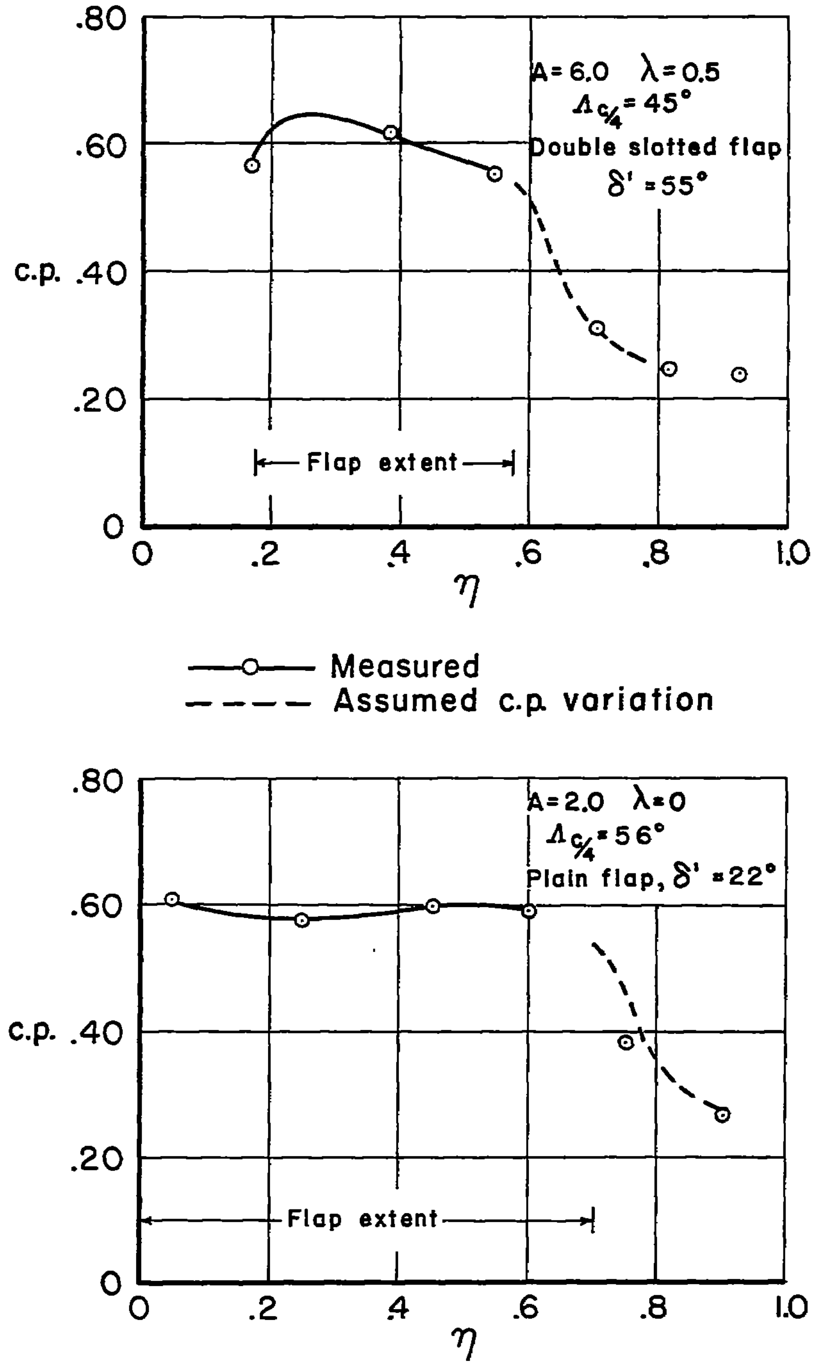


Figure 3.- Measured centers of pressure on two wings with partial-span flaps showing the assumed variation used in the transition region near ends of flaps.

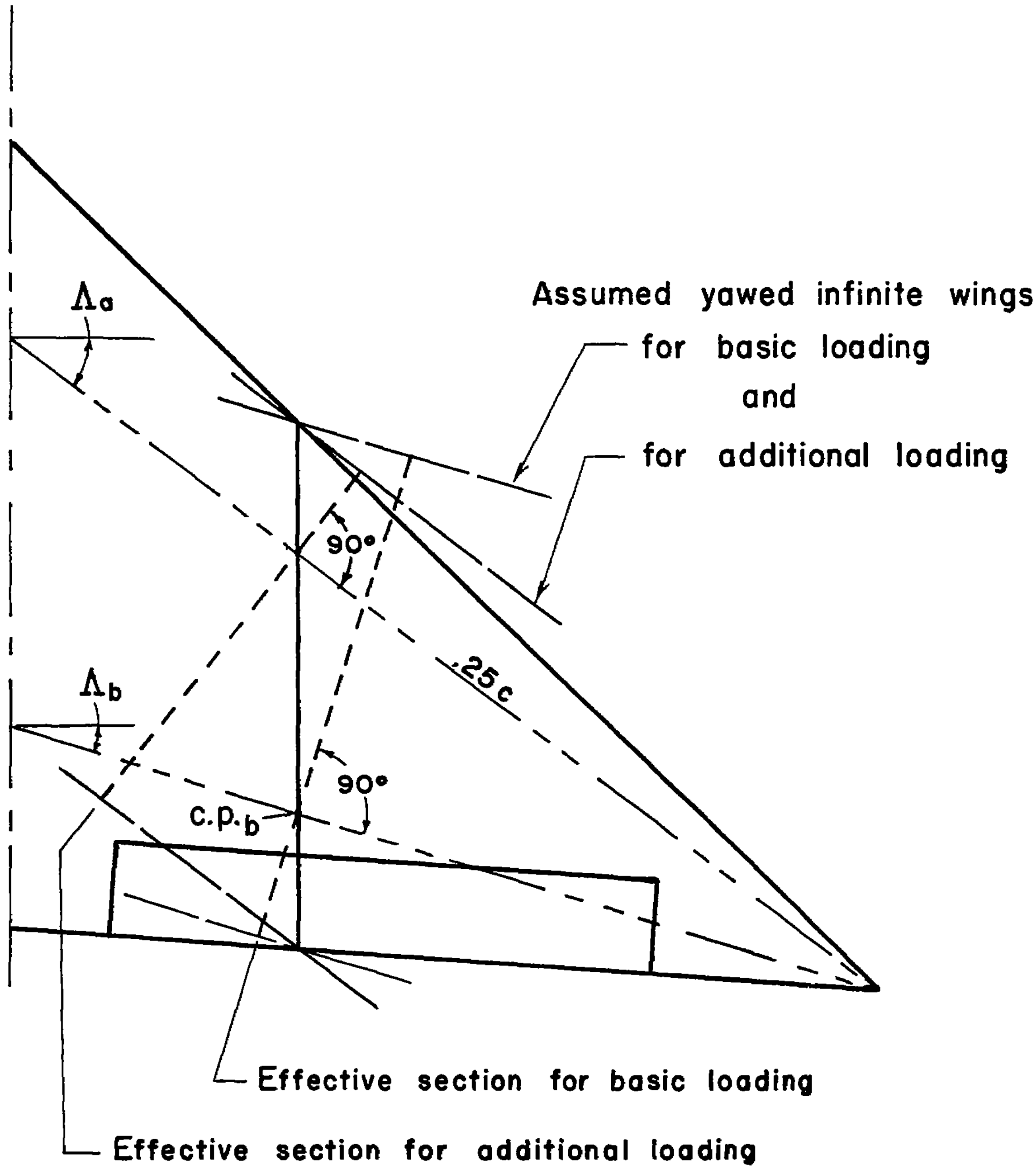
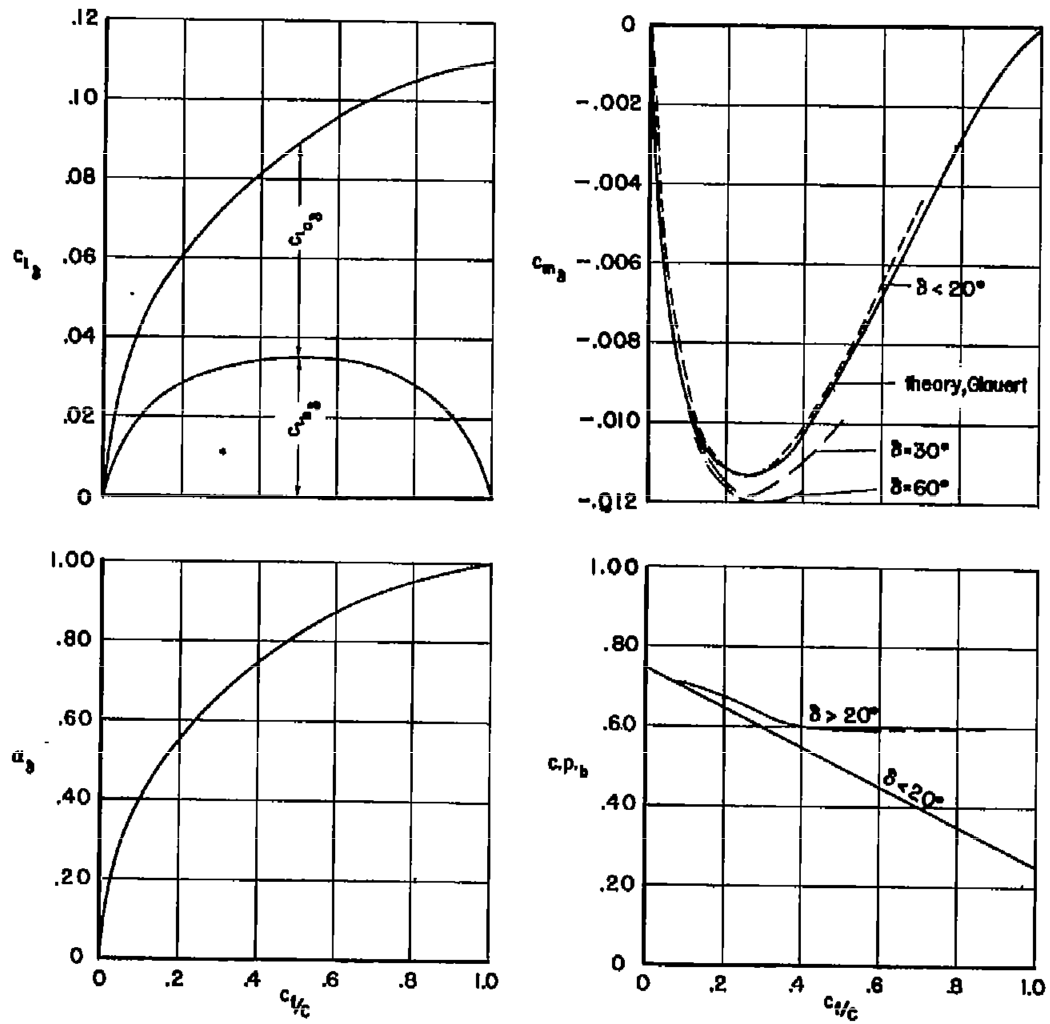
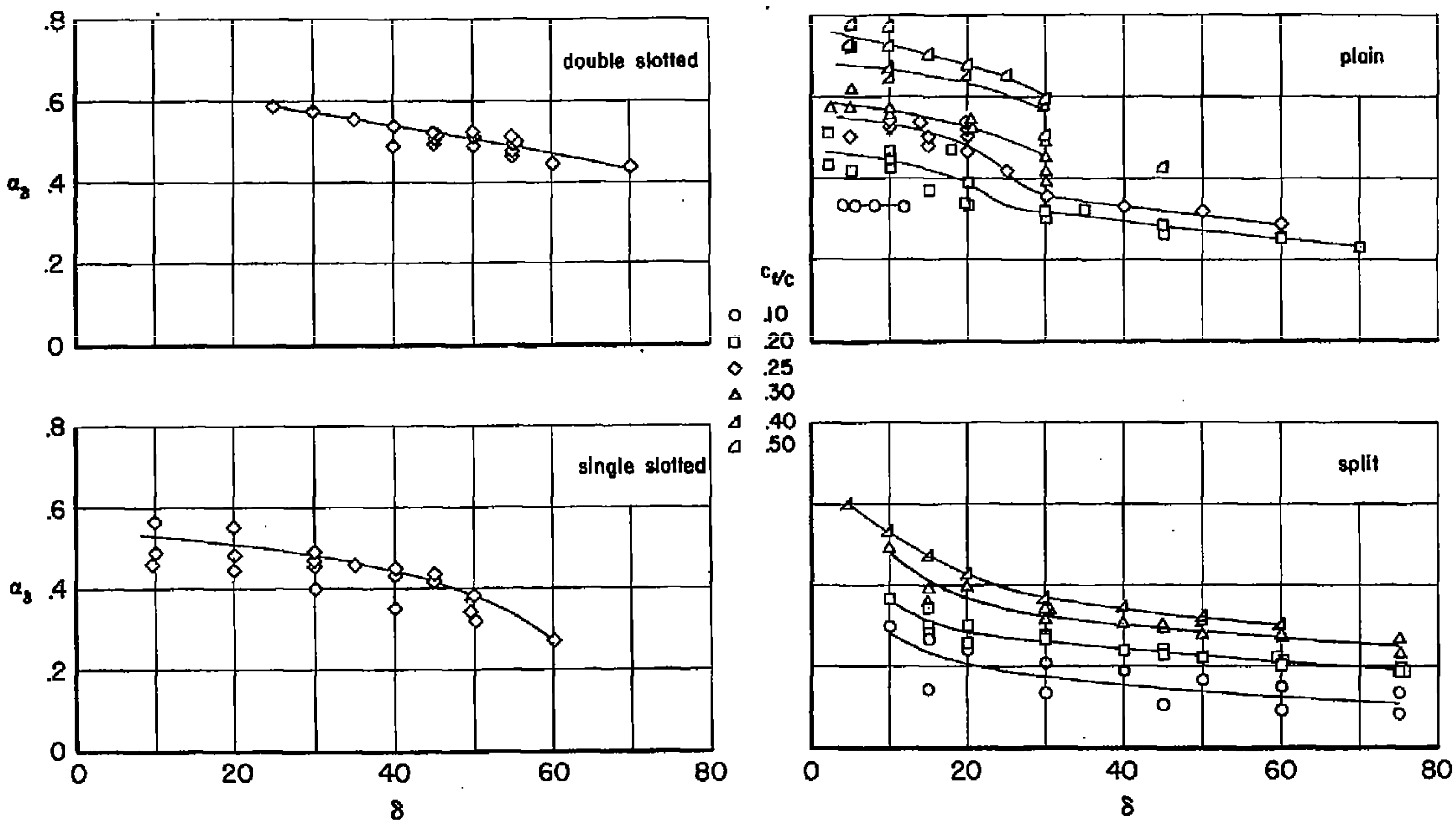


Figure 4.- Tapered swept wing with reference sweep lines shown for additional- and basic-type loading.



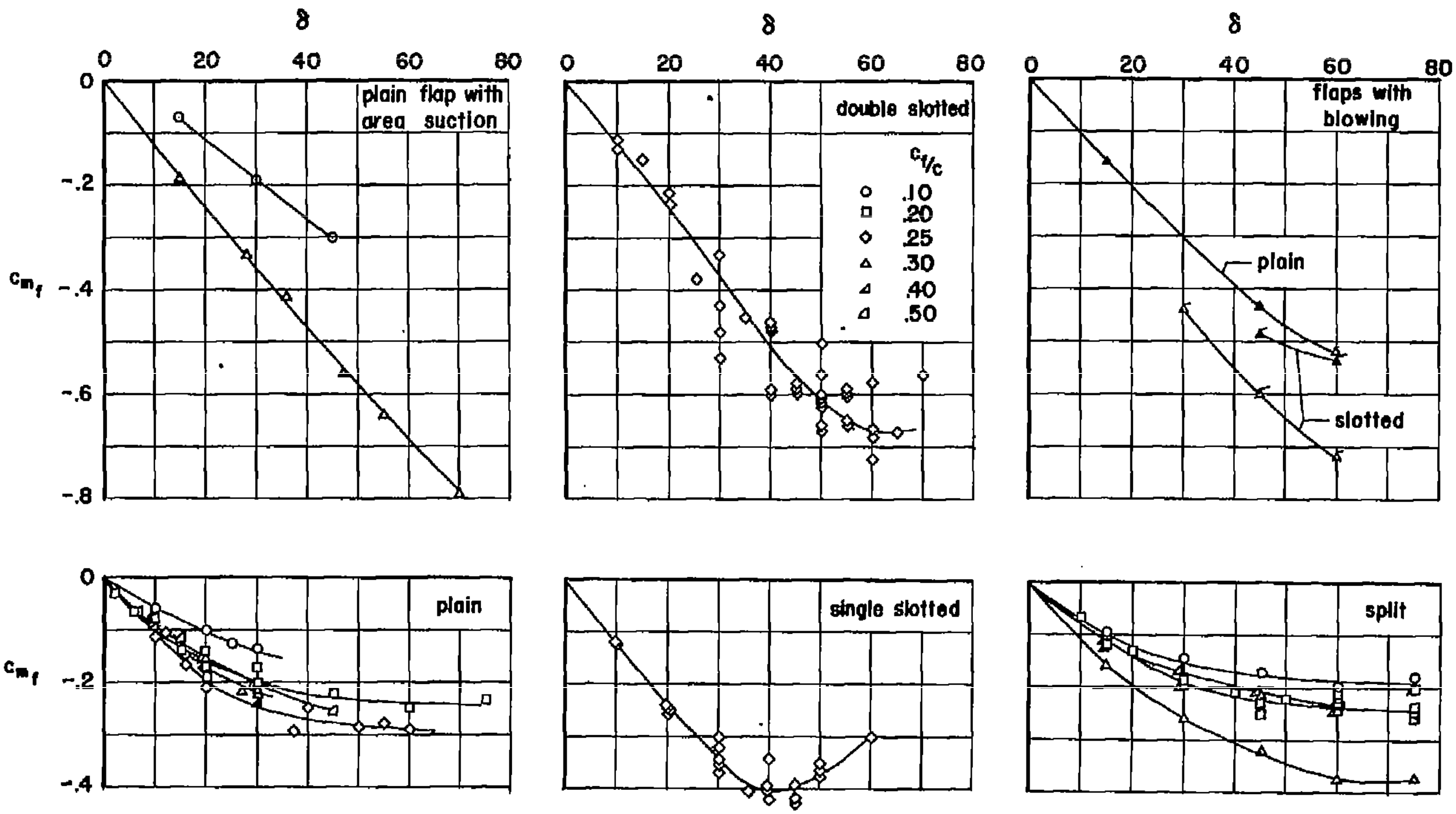
(a) Theoretical and calculated values for plain flap.

Figure 5.- Summary of two-dimensional-flap parameters.



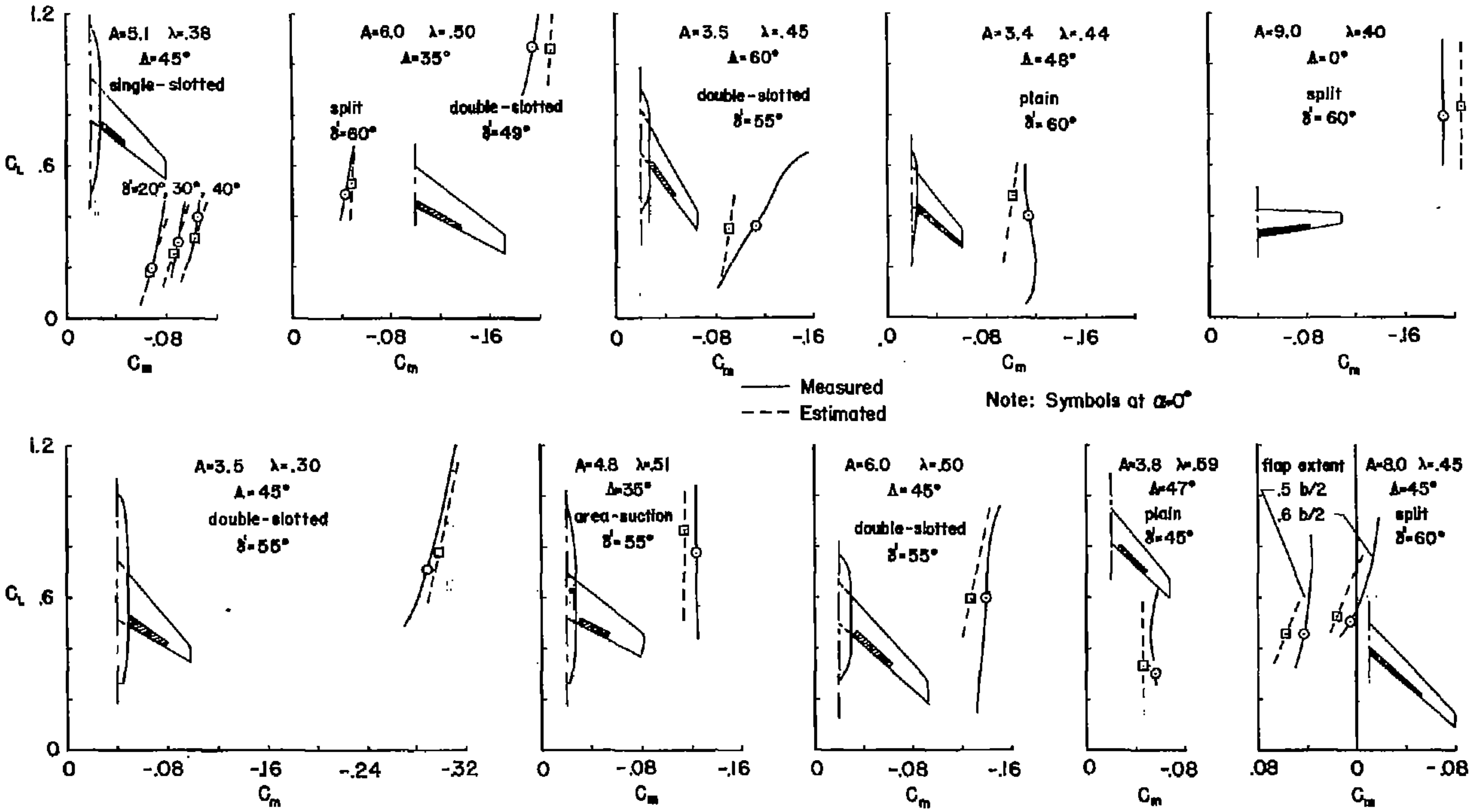
(b) α_δ based on two-dimensional test data

Figure 5.- Continued.



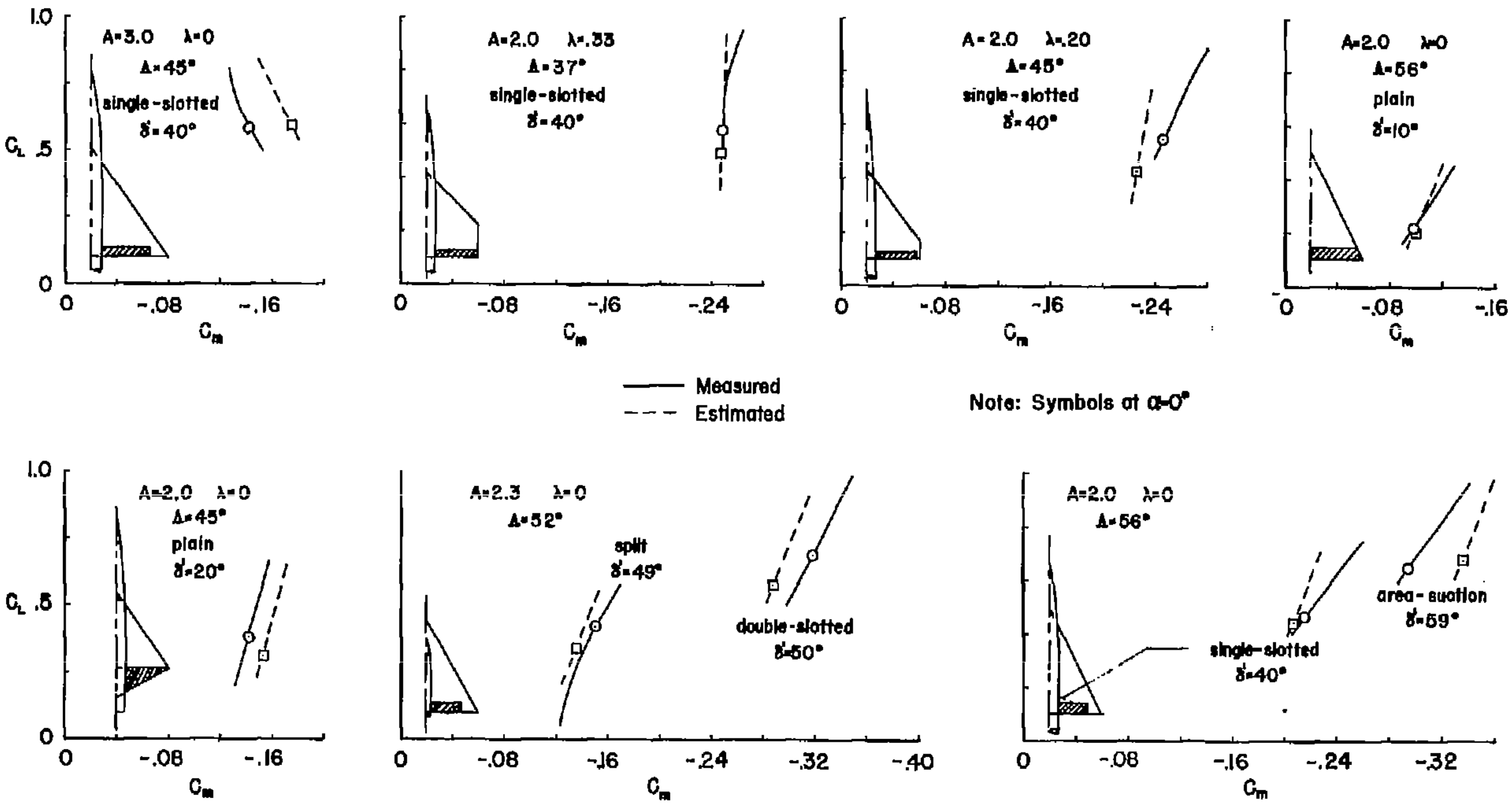
(c) c_{m_f} based on two-dimensional test data

Figure 5.- Concluded.



(a) A > 3, λ > 0.30

Figure 6.- Examples of the application of the method for estimating values of lift and pitching moment for α=0°.



(b) $A \leq 3$, $\lambda \leq 0.33$

Figure 6.- Concluded.

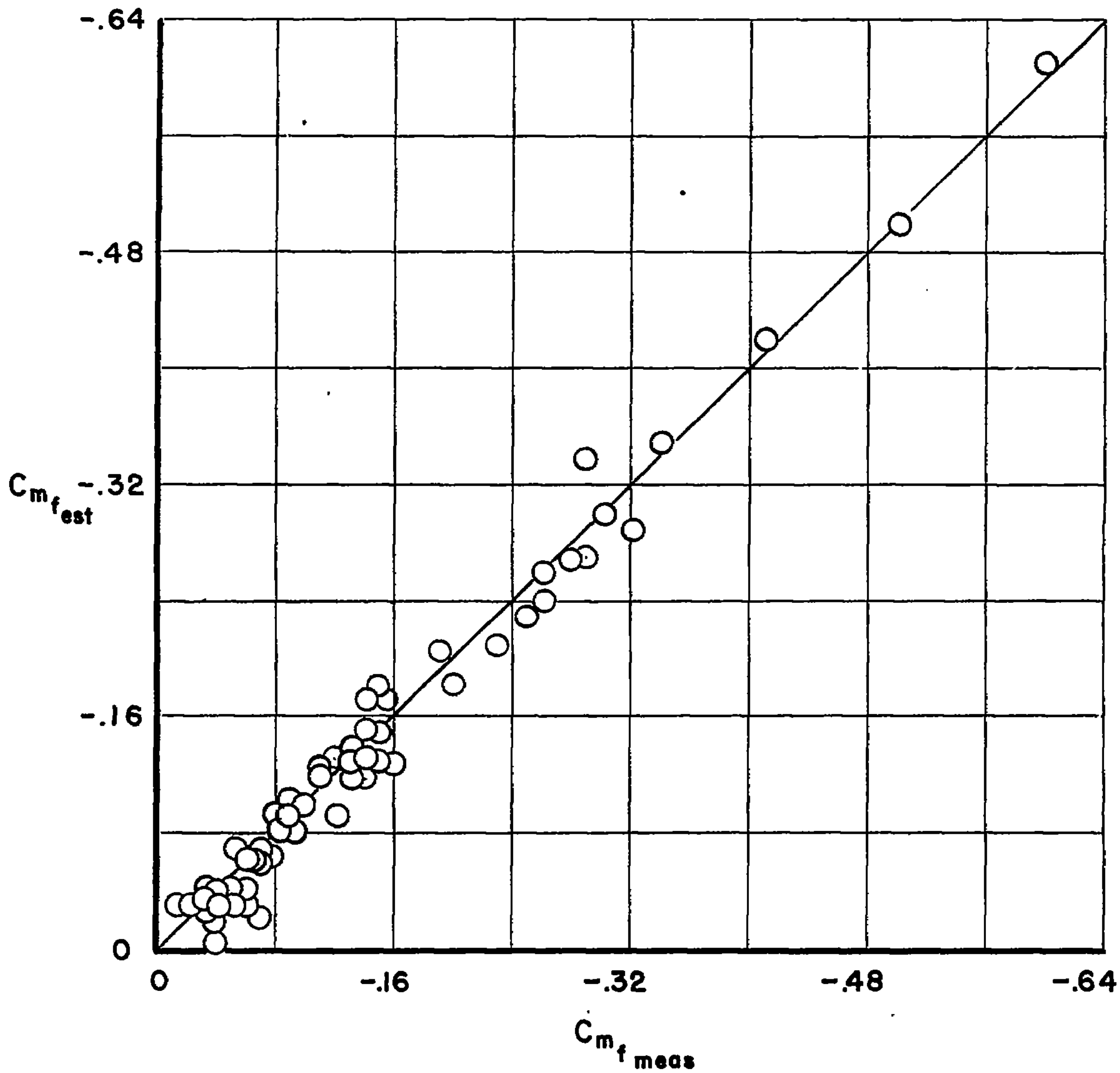


Figure 7.- Correlation of measured and estimated flap pitching-moment results for configurations in table I.

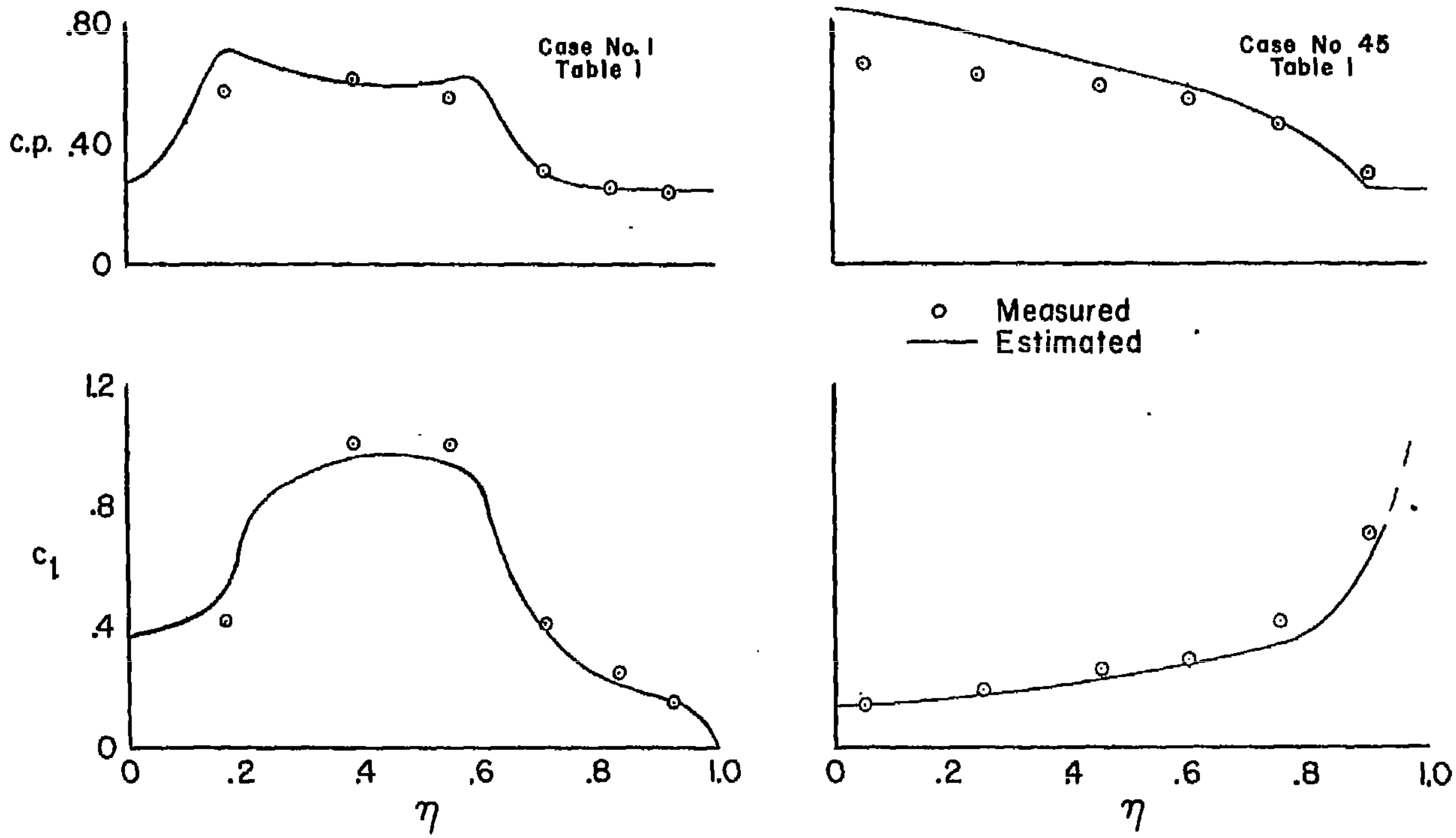


Figure 8.- Sample comparisons of measured and estimated span load distributions and local centers of pressure at $\alpha=0^{\circ}$ for a swept and triangular wing.

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